PREDICTION AND REDUCTION OF ROTOR BROADBAND NOISE

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INTRODUCTION

For helicopter rotors not undergoing bladeslap or other impulsive noise generation, broadband radiated noise is a dominant contributor to subjective weightings used to assess aircraft noise. Since the predominant trend in current and advanced helicopter designs is toward low speed rotors - i.e., those which avoid transonic relative tip Mach numbers at maximum cruise speeds, the relative need to control broadband noise will increase. There is a substantial body of research which has been performed on stationary and rotating airfoils, both for helicopter and non-helicopter applications, which can provide guidance in prediction and reduction of broadband rotor noise. This paper summarizes prediction techniques which can be or have been applied to subsonic rotors, and methods for designing helicopter rotors for reduced broadband noise generation. It is the primary purpose of this paper to show how detailed physical models of the noise source can be used to identify approaches to noise control.

SYMBOLS

B	number of rotor blades
c_0	speed of sound
C	numerical coefficient
$\tilde{C}_F$	force coefficient
D_r	dipole strength
f_k	frequency
F	force
G	Green's function
I	intensity
J_n	Bessel function

k	wavenumber variable
L	wetted length of trailing edge
ℓ_j (j=1,2,3)	turbulence correlation scale in the j-direction
M_o	characteristic flow Mach number U/c_o
M_{OR}	component of Mean M in the observer direction
M_V	V/c_o
M_0	U_0/c_o; also rotational Mach No.
p_I	pressure fluctuation without Kutta condition applied
q_0	mean dynamic pressure
r, R	distance from source to observer
t	thickness of wake
U	characteristic flow velocity
U_0	mean stream/flight speed
$\underset{\sim}{v}$	velocity
$\underset{\sim}{\bar{V}}$ or V_c	mean eddy convection velocity
υ	root mean square turbulence velocity
x_j (j=1,2,3)	rectangular coordinates
β	angle between V and x_1 -axis
Γ	vortex strength; also angle of gust with respect to the edge of airfoil
ρ, ρ_0	density
υ	numerical constant
Φ	spectrum function

ϕ observer orientation with respect to plane of rotation

ω, Ω radian frequency

MODELING CONSIDERATIONS

There are a variety of approaches to modeling aeroacoustic sources, ranging from exact mathematical descriptions typically involving solutions of Lighthill's acoustic analogy (1952), to semi-empirical descriptions of postulated sources using parametric data to arrive at prediction methods, to simple "data base" descriptions of special situations. Over the past 10 - 15 years, the state of rotor noise prediction capability has progressed from the "data base" stage to the exact solution stage, although many "exact solutions" require detailed physical measurements of flow field parameters which are beyond the current capabilities.

The modeling of broadband sources presented below will include representative available inputs from all levels of the above hierarchy. However, since our goal is to demonstrate that accurate source models, when stated in terms of measurable flow and geometric parameters, provide valuable insight into source reduction, we prefer to treat the broadband noise problem as a linear system analogy separating input and response functions.

To obtain a relationship between flow field characteristics and the sound radiated by lifting rotors, it is necessary to determine which mechanism is primarily responsible for the radiation. Each postulated mechanism leads to a particular characterization of the sound field and of its relationship to the flow field characteristics. Measurements and similarity arguments can then be used to establish *a posteriori* verification of the choice of the mechanism.

To form a clear physical view of the processes of interest in noise generation by rotating airfoils, it is desirable to analyze the various ways in which airfoils can interact with the surrounding viscous media to generate noise. The actual sources of sound generation by flow/surface interaction are fluid dynamic disturbances, which for the case of interest here may be separated into three categories, which may be separated as shown in Figure 1 (from Hayden, 1972, 1973).

Inflow turbulence, such as that arising from the atmosphere, previous blades, or the wake shear layer of an upstream disturbance, may produce lift and drag fluctuations of the whole surface.

A second category of fluid dynamic sources is the *turbulent boundary layer* which may radiate sound directly or, more importantly, creates a spanwise array of uncorrelated dipoles when it passes over the trailing edge. The characteristic dimensions of the boundary layer turbulence are small compared to the scale of the airfoil; therefore high frequencies are generally associated with turbulent boundary layer interaction with the trailing edge.

The third category of fluid dynamic sources is the *wake* of the airfoil, which may produce large scale "whole body" lift and drag fluctuations associated with large scale eddies in the wake, and "trailing-edge noise" at higher frequencies due to interaction (hydrodynamically) of small scale wake turbulence immediately downstream of the trailing edge with the edge. In the case of helicopter blades which follow each other in the same plane or in intersecting planes (e.g., tail rotors operating in main rotor downwash), the wake has further significance in that it becomes a source of inflow turbulence to downstream airfoils. The hydrodynamic wake and related acoustic spectra may be either very narrowband or broadband. Narrowband wakes are usually associated with laminar flow instabilities on an airfoil (or cylinder), whereas broadband wakes are usually associated with high Reynolds numbers wherein the upstream boundary layer is turbulent on both upper and lower surfaces.

Thus, the principal noise prediction problem becomes one of identifying and quantifying flow field parameters which "drive" the noise generation process.

The prediction of rotor broadband noise will now be addressed from the point of view of the source of excitation - environmental turbulence, versus self-generated boundary layer and wake turbulence.

It should be noted that much of the treatment of the broadband noise problem has been derived from work on stationary (non-rotating) airfoils, half planes, and the like. The utilization of the stationary surface information on rotating systems involves accounting for radial variation of mean and unsteady flow parameters, departures from quasi-two-dimensional flow situations at the tip of a rotating blade, and rotating acoustic source effects.

PREDICTION OF BLADE/TURBULENCE INTERACTION NOISE

Helicopter main rotors experience random fluctuations in the amplitude and direction of inflow due to atmospheric turbulence and, under some flight conditions, by turbulence from preceding rotor blades. Tail rotors operate in the main rotor downwash, atmospheric turbulences and, occasionally, in the wake of the fuselage and tail boom. A lifting rotor produces a mean downward velocity field which draws environmental eddies through the rotor plane with a convection velocity V_C. This random variation in "upwash" induces a random variation in the angle of incidence and hence a fluctuating blade load. Regardless of the source of inflow disturbance, the turbulence contains a spectrum of wave number components, resulting in a blade loading spectrum which produces acoustic energy over a wide range of frequencies.

Many of the possible mechanisms for broadband noise generation by a single stationary airfoil in a moving stream have been the subject of various investigations. The non-uniform velocity field associated with turbulence leads to an unsteady blade load that is dipole in nature. Recent thorough models of the problem include that of Amiet (1975), Homicz (1974), Amiet (1976), and Aravamudan and Harris (1978).

Formulation

Homicz and George (1974) gave an expression for the frequency f_0 beyond which the spectrum of the broadband noise is smooth as follows:

$$\frac{f_0}{\Omega} \simeq \frac{B\left(1 + \frac{M_0}{M_C}\right)}{2(1-M_0\cos\phi)} \tag{1}$$

where B is number of blades, M_0 and M_C are rotational and convection Mach numbers and ϕ is the observer orientation with respect to plane of rotation. This implies that at frequencies above f_0, the consecutive blade passage time is much greater than the eddy convection time and hence there is no significant blade-to-blade correlation. Under such circumstances, the total radiation may be obtained by summing the uncorrelated sound power spectral densities of each of the blades. The radiation of a single blade approximated by a rotating point dipole has been given by Ffowcs Williams and Hawkings (1969) as

$$\langle S_{pp}(r,f)\rangle = \frac{f^2}{4\rho_0 c_0^3 r^2} \sum_{n=-\infty}^{\infty} D_r(\phi, f-n\Omega) J_n^2\left(\frac{fR\cos\phi}{c_0}\right) \tag{2}$$

where $D_r(\phi,f)$ is the power spectral density of the dipole strength in the direction of radiation. Since the dipole strength is a direct consequence of the unsteady lift acting on the airfoil, it may be readily expressed in terms of the power spectral density of the unsteady lift

$$D_r(\phi,f) = \Phi_{LL}(f)\ \sin^2\phi \tag{3}$$

Under the assumptions of stationary and homogeneous random processes, the unsteady lift response function is related to excitation spectral density (turbulent upwash) and the aerodynamic transfer function in the following manner:

$$\Phi_{LL}(f) = \iiiint_{-\infty}^{\infty} d^3k\,d\tau\,\Phi_{ww}(\vec{k})\ |K(k_c,k_s)|\ \exp[i2\pi(k_xQ - f)\tau] \tag{4}$$

which may be simplified to

$$\Phi_{LL}(f) = \frac{2}{Q}\iint_{-\infty}^{\infty} dk_y dk_z\ \Phi_{ww}(f/Q,k_y,k_z)\ |K(f/Q,k_y)|^2 \tag{5}$$

where Φ_{ww} is the spectrum of turbulence upwash and K is the aerodynamic transfer function. Equation (5) is general in the sense that any known turbulence upwash spectrum and/or an aerodynamic transfer function may be used to yield appropriate lift power spectral densities.

Effects of Aerodynamic Transfer Function

If we assume that at every instant the rotor blade behaves like a two-dimensional airfoil in a three-dimensional sinusoidal upwash pattern, the three governing parameters for the aerodynamic transfer function are k_t, Γ and M_0. Here

$$\Gamma = \tan^{-1}\left(\frac{k_s}{k_c}\right) \text{ and } k_t = (k_c{}^2 + k_s{}^2)^{\frac{1}{2}}$$

where k_c and k_s are components of the gust wave vector parallel to the chord and span of the blade, respectively, and are normalized with respect to blade semi-chord. During each revolution, a blade will encounter some regions in which Γ approaches a right angle, implying nearly steady but definitely three-dimensional flow. In some other region, Γ may be very small, so that the flow is nearly two-dimensional, but definitely unsteady. In either case, the loading is less than that predicted by quasi-steady two-dimensional theory.

To check this quantitatively, Homicz and George (1974) computed the load response to a convected sinusoidal upwash pattern of amplitude w_0 and wave number k_t for the two extreme cases of $\Gamma = 0$ and $\Gamma = \pi/2$ and various Mach numbers. The results for $M_0 = 0.4$ are presented in Figure 2. The curve for $\Gamma = 0$ was obtained from Osborne's (1973) asymptotic expression for the compressible extension of the two-dimensional Sears function (1941):

$$|S_{eff}| = \left[J_0^2 \left(\frac{M_0^2 k_t}{\beta^2}\right) + J_1^2 \left(\frac{M_0^2 k_t}{\beta^2}\right) \right]^{\frac{1}{2}} \left| H\left(\frac{k_t}{\beta^2}\right) \right| \qquad (6)$$

where $\beta^2 = (1 - M_0^2)$ and $H(k)$ is the magnitude of the original incompressible Sears function which, to a very good approximation, is given by

$$H(k) \simeq (1 + 2\pi k)^{-\frac{1}{2}}$$

The $\Gamma = \pi/2$ curve in Figure 2 was obtained from Filotas' (1969) work on the $(k_t, \Gamma, 0)$ solution in conjunction with a similarity rule derived by Graham (1970). From Figure 2, it is evident that the two curves are rather close at low frequencies, tend to diverge as frequency increases, and approach the same slope at high frequencies. This result is of significance in discussing the effect of swept blades on noise reduction, where the value of Γ is ranging from 0 to approximately $\pi/2$ along the span of the blade, thus reducing the unsteady lift fluctuations.

Effect of Free Stream Turbulence

For a given unsteady aerodynamic transfer function, the unsteady lift response function is dependent only on the spectrum of turbulent upwash. While considering the response of main rotors to atmospheric turbulence, Homicz and George (1974) used the Dryden form of the spectrum for turbulence input:

$$\Phi_{ww}(k) = 64\pi^3 \bar{\omega}^2 \Lambda_f^5 \frac{k_x^2 + k_y^2}{[1+4\pi^2 \Lambda_f^2 k^2]^3} \qquad (7)$$

where $k^2 = k_x^2 + k_y^2 + k_z^2$ and Λ_f is the integral scale of turbulence. Their predicted sound pressure levels were lower than the measured full scale rotor spectra of Leverton (1973). Aravamudan and Harris (1978) used a von Karman spectrum of turbulence and compared the computed results with the measured sound pressure levels of a model rotor operating in an open jet anechoic wind tunnel facility. Figure 3 shows a comparison of their measured and computed results for two Mach numbers and integral scales of turbulence. The details of these measurements and predictions are presented in another paper by Aravamudan, Lee and Harris (1978) in this conference.

The resulting scaling law to scale blade/turbulence interaction noise from model tests is:

$$SPL_p = SPL_m + 20 \log \frac{b_p c_p}{b_m c_m} + 10 \log \frac{\bar{\omega}_p^2}{\bar{\omega}_m^2}$$

$$+ 40 \log \frac{M_{tp}}{M_{tm}} + 20 \log \left[\frac{(1+\mu_p)^2 + \left(\frac{M_{cp}}{M_{tp}}\right)^2}{(1+\mu_m)^2 + \left(\frac{M_{cm}}{M_{tm}}\right)^2} \right]$$

$$- 3.3 \log \frac{\Lambda_{fp}}{\Lambda_{fm}} - 20 \log \frac{r_p}{r_m} + 10 \log \frac{\sin^2\phi_p}{\sin^2\phi_m} \qquad (8)$$

where subscripts p and m stand for prototype and model rotors, b is the blade span, c is blade chord, μ is advance ratio, M_c is convection Mach number through rotor disc, and M_t is tip Mach number.

To facilitate comparison with the full scale rotor data of Leverton, we used the prediction procedure developed in Aravamudan et al. (1978) with the estimated turbulence properties of $\Lambda_f = 0.57$m, $(\bar{\omega})^{\frac{1}{2}} = 1$ m/sec (George and Kim (1976)). The resulting sound pressure spectrum is compared with the measured spectrum of Leverton (1973) in Figure 4. Similar analyses can be used for the prediction of broadband noise radiation from tail rotors operating in the downwash of main rotors; however, the appropriate aerodynamic inputs have not been measured in detail.

TRAILING EDGE NOISE

When turbulent boundary layers or other surface-attached air flows pass over a discontinuity in the surface such as a trailing edge, substantial sound may be radiated. In the last ten years, there has been a proliferation of theoretical and experimental work aimed at describing the sound generation by this fluid dynamic source for a semi-infinite half plane. Following on the suggestions of Powell (1959), Hayden (1969) and Chanaud (1970) attempted to model the trailing edge sound generation process as a distribution of point dipoles, and developed some semi-empirical correlations with data from a wall jet over a rigid plate. At the same time, Ffowcs Williams and Hall (1970) were solving Lighthill's acoustic analogy for a semi-infinite half-plane with a generalized turbulent flow past the edge. Since that time, several models have been advanced by numerous authors. Unfortunately, many of these models have been based on different flow field approximations and different surface/observer geometries. Therefore a direct comparison of the models was difficult. Recently, Howe (1977) undertook a comparison of various half-plane models for a situation of consistent flow field conditions. He found that the predominant source mechanism is dipole in nature and that, when cast in terms of a common system of flow parameters for a semi-infinite plane, most of the theories predicted the same result for low Mach number conditions:(Ffowcs Williams and Hall (1970); Crighton (1972); Chandiramani (1974); Chase (1972, 1975); Hayden et al. (1976); and Howe (1977)). Howe developed a unified model which also took into account moving medium and moving source effects. The details of this model are contained in Howe (1977). For the half-plane geometry shown in Figure 5 , Howe's model predicts the following relationship between flow parameters and radiated sound for "blade-fixed" coordinates:

$$\langle p_I^2 \rangle = \frac{C\rho^2 v^2 V^2 M_V}{2\pi} \left(\frac{L\ell^3}{R^2}\right) \frac{\sin\alpha \sin^2\left(\frac{\bar{\theta}}{2}\right)\cos^3\beta}{(1+M_{OR})^2(1-M_{VR})^2(1-M_{V1}\sin\alpha)} \qquad (9)$$

Howe's work concluded that the question of application of the Kutta condition at the trailing edge is unresolved with substantial differences occurring in predicted levels depending upon the presence or absence of this condition. Furthermore, for finite chord surfaces, the relationship is expected to become more complicated; indeed, the trailing edge noise problem has not been fully described analytically for finite surface geometries. Several parameters in Howe's equation are of interest from the point of view of reduction of broadband noise. These are turbulence velocity v, mean and eddy convection velocities, spanwise length scale, and the angle β, which is the angle between the trailing edge and the direction of the turbulent flow. From the point of view of reduction of trailing edge noise, one might seek to modify these parameters by either selecting a location for the rotor where the turbulence parameters are minimized, or by modifying the geometry of the rotor to take advantage of the sweep effects (e.g., $\cos^3\beta$). Furthermore, surface modifications (e.g., leading edge serrations or vortex generators) could, in principle, be used to modify the turbulence intensity and length scale parameters. Application of these concepts will be discussed in subsequent sections.

"VORTEX SHEDDING" NOISE FROM ROTATING AIRFOILS

It has been known since the early 1900's that the shedding of a vortex wake produces sound, and there is a rather large literature on this subject. Much of this literature is concerned with the frequency of vortex shedding and of the attendant sound, for a variety of two-dimensional shapes. More recently Hanson (1970), Hersh and Hayden (1971), and Patterson *et al.* (1973) made studies of vortex shedding noise from isolated stationary and rotating airfoils and arrived at different expressions to predict the frequency of discrete tones. Hanson used a wake momentum thickness, Hersh and Hayden used a wake thickness, and Patterson *et al.* found good correlation by using a laminar boundary layer thickness on the pressure side of the airfoil. Aravamudan, Lee and Harris (1978) performed a series of experiments with a model rotor and found that a Strouhal scaling with respect to thickness of the airfoil yielded favorable comparisons with measurement. However, since all of the shear layer thicknesses are interrelated for unstalled airfoils, the frequency prediction issue has become a second-order affair.

For cases when the frequency of unsteady vorticity "shedding" is such that the resultant acoustic wavelength is much greater than the chord of the airfoil, the unsteady force can be represented as an acoustic dipole, as first noted by Yudin (1947). Later, Phillips (1956) showed by starting with Curle's equation that, for low Mach numbers, and for the case of an acoustically-compact rigid body experiencing a harmonic force, the radiated acoustic intensity at distance r is:

$$I = \frac{9f^2|F_0|^2\cos^2\theta}{8\rho c_0^3 r^2} \tag{10}$$

Nondimensionalizing F with a force coefficient leads to

$$F = \frac{1}{\sqrt{2}}\rho U_0^2 \quad t\ell_3 \ \tilde{C}_F \tag{11}$$

where $\tilde{C}_F$ is the rms oscillating lift coefficient, t is the thickness, and ℓ_3 is a spanwise length over which the shed vortices are correlated, which may be small compared to the total span of a blade. Integrating the resulting expression for the intensity of sound over the span of a rotating blade leads to the total acoustic intensity which may be scaled as follows:

$$I \simeq \left(\frac{ft}{U_0}\right)^2 \tilde{C}_F^{\,2} \; M_0^3 \, U_0^3 \, b\ell_3 \; \cos^2\theta \tag{12}$$

where b is the span of the rotating blade and θ is the angle between the fluctuating force axis and the observer. Since rms fluctuating lift is, in general, a function of strength of shed vortices and the separation of vortex sheets, it can also be represented in terms of the steady state drag coefficient C_D. Noting that a Strouhal number can be defined by:

$$S = \frac{ft}{U_0} \tag{13}$$

this leads to an expression for the total acoustic intensity as follows:

$$I \simeq S^2 C_D^{\,2} \left(\frac{\tilde{C}_F}{C_D}\right)^2 \; M_0^3 \, U_0^2 \, b\ell_3 \; \cos^2\theta \tag{14}$$

Ross (1964; see also Ungar *et al.*, (1972)) showed that expressions for the dimensionless drag coefficient and Strouhal number have reciprocal dependencies on the ratio of wake separation h, to the body thickness t. As a result, the product is independent of the relative wake width. The resultant expression is dependent only on the relative induced velocity u/U_0. The relative induced velocity is a function of the shape of the body shedding vorticity, and tends to be constant for a given shape over a wide range of Reynolds number. As discussed in detail in the literature survey by Ross (1964), experiments show a clear trend for the oscillating lift to be a constant fraction of the drag coefficient over a wide range of Reynolds numbers. For cylinders, this fraction is about one-third and there are no data available for other shapes. As we are dealing with the drag associated with the shed vortices, it would follow that for other shapes the rms oscillating lift will also be approximately one-third of the steady-state profile drag. The unknown function ℓ_3, the spanwise coherence length, is the only expression that is strongly dependent on the Reynolds number which is essential in explaining some of the observed phenomena.

Thus the key issue in predicting "vortex noise" is estimating wake intensities and spanwise correlation lengths. Aravamudan, Lee and Harris (1978) present further discussion on this matter.

It should also be noted that there are several empirical methods available to predict the high frequency broadband noise radiation from helicopters rotors and propellers. Ungar *et al.* (1972) reviewed the existing literature and obtained the following expressions for the overall sound pressure levels (SPL) due to vortex shedding from airfoils:

$$SPL = -48 + 10 \log A_b + 60 \log U_{0.7} + 10 \log\left(\frac{C_D}{0.01}\right)$$

$$+ 10 \log \{\tfrac{1}{2}[1-J_0(2\beta)\cos 2\theta]\} - 20 \log \frac{r}{3} \quad (15)$$

where A_b is the disc area, $U_{0.7}$ is the rotational velocity at 70% of blade span and β is pitch angle. The spectrum was calculated using the Strouhal frequency relation

$$f_p = 0.28 \frac{U_{0.7}}{d_p} \quad (16)$$

where $d_p = t \cos \alpha + c \sin \alpha$ and is the wake-projected airfoil thickness. In Figure 6 is shown the comparison between predicted and measured vortex noise spectra for the H-D 1 hovercraft using equations (15) and (16).

NOISE REDUCTION AT THE SOURCE

The preceding formulations for the various flow/surface interaction noise mechanisms can be examined to identify physical parameters which could, in principle, be altered to reduce the strength of the noise source, or change its characteristic frequency. In general, the approaches can be roughly reduced to two classes of options:

- reduce the aerodynamic excitation, or
- reduce the response to the excitation.

Below we examine such approaches to reducing the broadband noise generation for the blade/turbulence interaction source and the trailing edge sources, including "vortex shedding". Since it has long been obvious that all rotor sources can be reduced by reducing the rotation speed, the following discussion presupposes that designers would take advantage of lower rotor speeds to the maximum extent allowed by performance requirements, Thus the approaches presented could be viewed as a means for additional noise reduction, or alternatives to further lowering of rotor speed.

Blade/Turbulence Interaction Noise

The best approach to reduction of blade/turbulence interaction noise is to try to locate the rotor blades where inflow turbulence will be minimized. On single rotor helicopters, the primary source of inflow turbulence to the main rotor is the atmosphere, except in certain flight modes where turbulence from preceding blades may be encountered. Fortunately, the most intense atmospheric turbulence occurs at such low frequencies that its farfield acoustic consequences are negligible.

Tail rotors often experience a mean inflow distortion and turbulence which are created by the downwash from the main rotor, in conditions of forward flight. Both of these conditions can be mitigated in the initial design stages of the helicopter by locating the tail rotor outside the envelope of main rotor downwash trajectories expected throughout the flight regime. Pegg and Shidler (1978) discuss the results of an extensive experimental program which demonstrated this most powerful approach for reducing tail rotor noise.

If one is unable to avoid inflow turbulence to a particular rotor, then the only available approach to reducing the noise is to reduce the fluctuating loads experienced by the blades operating in the turbulent flow. If the predominant reduced frequency is high enough, increasing the blade chord could reduce the response; however, substantial increases are required to achieve much benefit from this effect, and other aerodynamic design tradeoffs would be required. An interesting and promising technique has been advanced by Brown *et al.* (1977), which is based upon the premise that the component of inflow velocity (both mean and turbulent) which causes the blade response is that normal to the leading edge. Using a dipole model for the noise generation process, they determined that the radiated intensity is related to the flow velocity by $\cos^3\beta$ where β is the angle between the flow and the vector normal to the leading edge. (Note that Howe (1978) also finds the same result for turbulent boundary layer/trailing edge interaction noise (Eq. 9)). Furthermore, the frequency of the noise generation will vary with the cosine of β. Thus, the hypothesis is made that propellers with their leading edges swept forward or backward will generate less blade/turbulence interaction noise than a blade with radial leading edge. Brown tested two 40 cm dia. propellers in the BBN acoustic wind tunnel to verify this hypothesis - one with a symmetrical chord distribution about a radial line, and the other with 51° of midchord sweep. The chord lengths at corresponding radii were identical. A consistent source of broadband turbulence was introduced upstream of the propellers, and a range of advance ratios was explored. The results shown in Figure 7 are quite dramatic and agree well with what would be predicted by the $\cos^3\beta$ intensity model, and frequency shifting by $\cos\beta$.

We note at this time that swept blades have been successfully used to reduce noise due to steady loads and thickness effects on a high speed turbofan (Hayden *et al.*, 1977) and that Farassat (1978) has recently predicted the same trends for high speed free rotors (see paper by Farassat, Nystrom, and Brown in this conference). Thus, the swept blade concept for reducing broadband noise is compatible with a demonstrated concept for reducing discrete frequency noise.

Trailing Edge and "Vortex Shedding" Noise Reduction

Trailing edge sources can also be reduced by modifying turbulence parameters (intensity and length scales), or by sweeping the trailing edge relative to the mean flow direction. In helicopter rotors, excessive local load levels often occur, leading to intense local blade-generated turbulence, and consequently high broadband noise levels. Thus, a high priority is to attempt to design rotors to avoid excessive local loading. Given the overall complexity of the helicopter's aerodynamics and the general lack of available data on full-scale helicopters, the ability of designers to avoid local blade stall in all flight regimes may be limited for some time.

Modification of Turbulent Structure

In unstalled portions of a rotor, the turbulence structure can, in principle, be modified to reduce trailing edge sources, by the use of vortex generators or leading edge serrations. The latter technique has been studied as a means for reducing the so-called high frequency broadband noise peak in propellers and rotor noise spectra (see, for example, Hersh and Hayden (1971), Hersh *et al.*, (1974)). Figure 8 shows the noise reduction achieved on a model rotor for varying spanwise amounts of leading edge treatments. Although the high frequency peak is generally thought to be associated with pressure-surface laminar flow instability, and therefore possibly limited to small low speed rotors, George (1977) and Aravamudan *et al.* (1978) point out the existence of this peak in full scale rotor data. Thus, the serration concept could be useful in reducing this source.

Sweep

Arguments similar to those made above regarding the importance of the normal velocity component in noise generation can be advanced for trailing edge mechanisms. Howe's formulation for semi-infinite surfaces predicts a $\cos^3\beta$ dependence on overall radiated intensity. A dipole model for wake vortex shedding noise such as discussed earlier would predict a $\cos^5\beta$ intensity dependence (e.g., see Brown, 1977). Both models predict characteristic frequency variation with $\cos\beta$. The results of an experiment conceived to demonstrate this effect are shown in Figure 9. One propeller had a radial trailing edge while the other had a smoothly curved trailing edge which always produced a normal velocity vector about 45° from radial. The results seem to indicate that the $\cos^5\beta$ law is closely followed.

Porous Trailing Edge

The final concept to be discussed for reduction of broadband noise is the porous or variable impedance trailing edge concept, which involves replacing the solid surface of the blade with a surface(s) which is (are) porous, thus allowing a more gradual acceleration of the medium around the edge than occurs when boundary layer turbulence encounters a sharp surface discontinuity. Further, the porous surfaces near the edge reduce the intensity of various band wakes, since the strong pressure gradients involved in the formation of such wakes cannot exist. Further background on the concept is given in Hayden and Chanaud (1973 a,b) and in Hayden (1976). Figure 10a illustrates the notion that there exists a variety of impedance-change contours which can serve to reduce the noise generation. Figure 10b indicates that there are at least four general structural arrangements possible to implement the concept, all of which have been investigated on stationary surfaces. There is presently analytical work under way at BBN to develop methods for optimizing the reduction of a single wavenumber component, or the entire spectrum, based upon surface porosity, edge length/wavelength, etc.

Some results pertinent to helicopter rotors are given in Figures 11a and 11b, in which the nearly total elimination of the wake-related peak and 5 - 10 dB reduction of the broadband peak may be seen. Similar results have been measured on powered lift flaps and simple rotating blades (Hayden 1976;1977). We believe that the potential for 10 dB of broadband noise reduction exists for practical rotor systems. However, work on materials characterization and verification of analytical predictions is needed, as well as on the aerodynamics of airfoils with porous surfaces, with or without cavities behind them.

CONCLUSIONS

This paper has provided some evidence that accurate detailed source models cast in terms of physically measurable or controllable parameters can point the way to noise reduction approaches. Rotor broadband mechansisms are quite well understood for Mach numbers below the transonic regime; at higher speeds, additional mechanisms such as randomly-modulated thickness noise, shock/turbulence interaction, and shock instability may become important. These additional mechanisms are not presently well understood and deserve attention. *However, for all mechanisms, a critical deficiency exists in the specification of both mean and unsteady flow field components needed to estimate the strength, spectral characteristics, and spatial location of the noise mechanisms.* The effective selection and application of rotor noise reduction measures depends upon the ability to predict or measure the location of the source to be reduced. Further improvements in the rotor noise prediction area will come about only after development of a data base and prediction capability for details of steady and unsteady flows around rotors.

We have shown several promising noise reduction concepts that have been developed from, or can be explained by, accurate source models. The concepts have been verified on simplified systems for which important flow field information could be developed. The applicability to helicopters is clear, but the success in optimally implementing the concepts will depend upon the ability to develop the relevant design information, especially flow field conditions.

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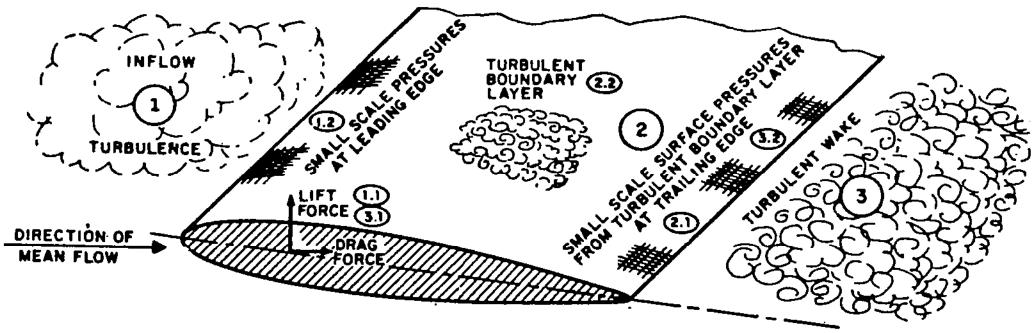


FLUID SOURCE	MAJOR ACOUSTIC SIGNIFICANCE
(1) INFLOW TURBULENCE	(1.1) LIFT AND DRAG FLUCTUATIONS OF WHOLE SURFACE (1.2) SMALL SCALE PRESSURE FLUCTUATIONS AT LEADING EDGE
(2) TURBULENT BOUNDARY LAYER	(2.1) SMALL SCALE PRESSURE FLUCTUATIONS AT TRAILING EDGE (2.2) ACOUSTICALLY "FAST" PRESSURES THAT RADIATE DIRECTLY
(3) TURBULENT WAKE	(3.1) LIFT AND DRAG FLUCTUATIONS ON WHOLE SURFACE (3.2) SMALL SCALE PRESSURE FLUCTUATIONS AT TRAILING EDGE

Figure 1.- Fluid dynamic sources of broadband noise.

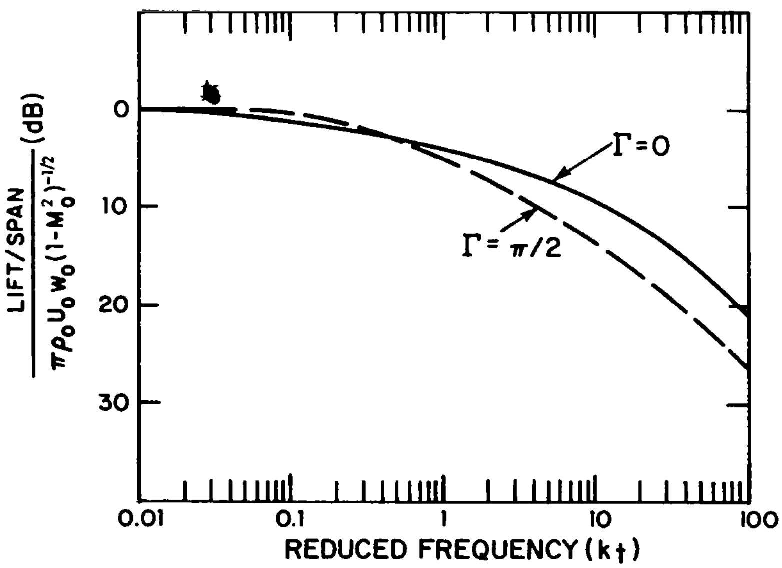


Figure 2.- Effect of spanwise component of gust velocity on lift response function.

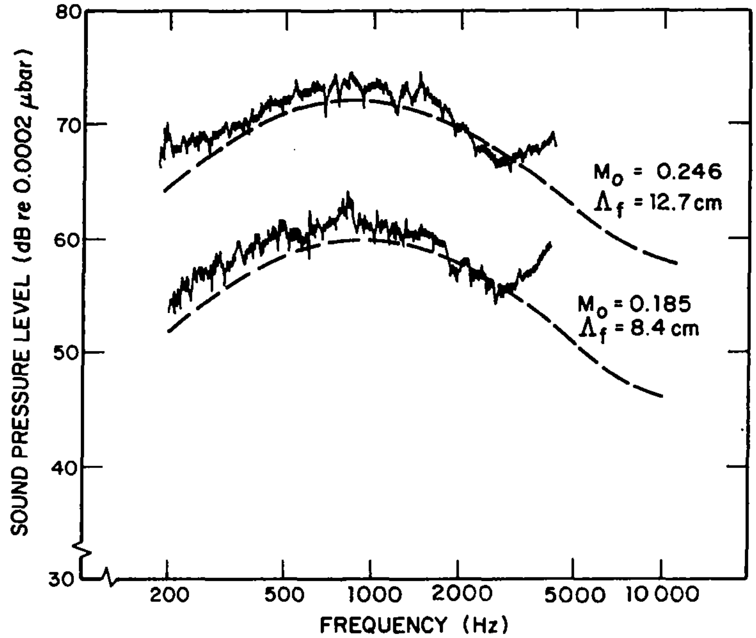


Figure 3.- Computed and measured spectra from model rotor in wind tunnel with simulated turbulence.

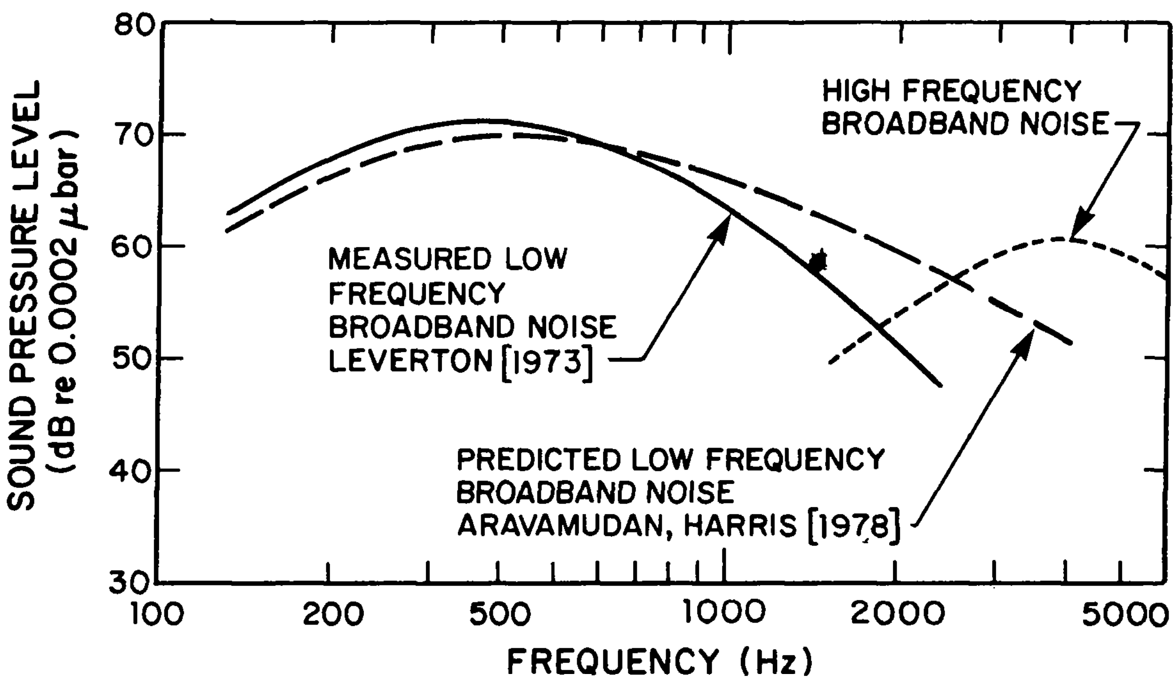


Figure 4.- Prediction of full-scale rotor broadband noise from model data.

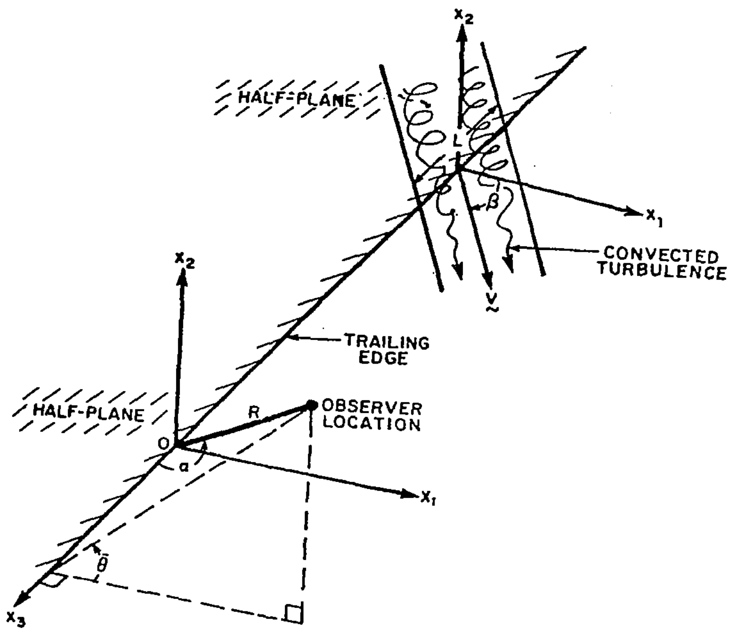


Figure 5.- Coordinate and flow field definitions for trailing edge noise calculations.

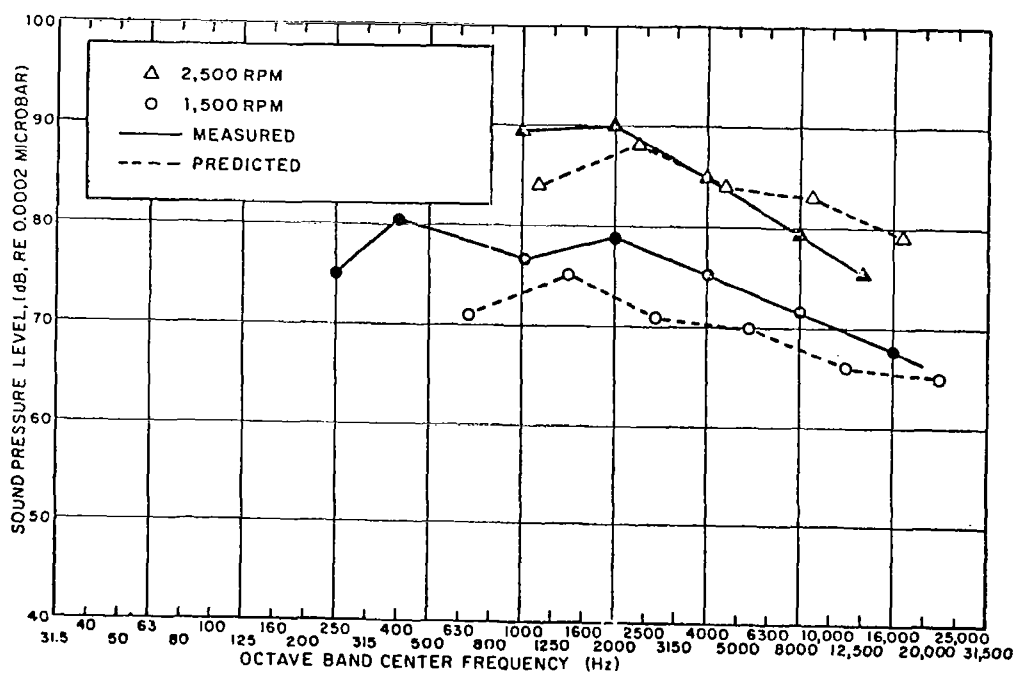


Figure 6.- Comparison of measured and predicted vortex noise for H.D. 1 Hovercraft at 15° out of propeller plane, 30 m from hub.

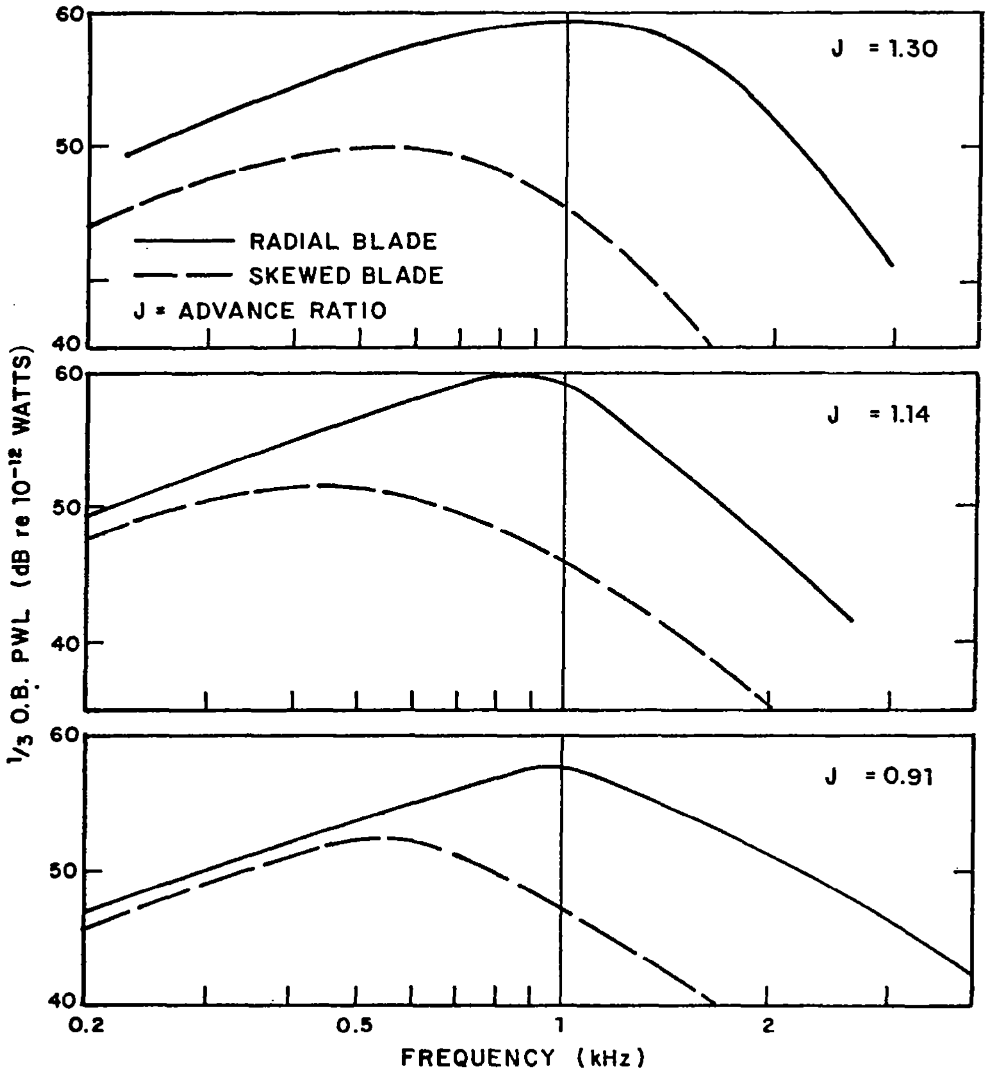


Figure 7.- Effect of 50° sweep on rotor/turbulence interaction noise spectra.

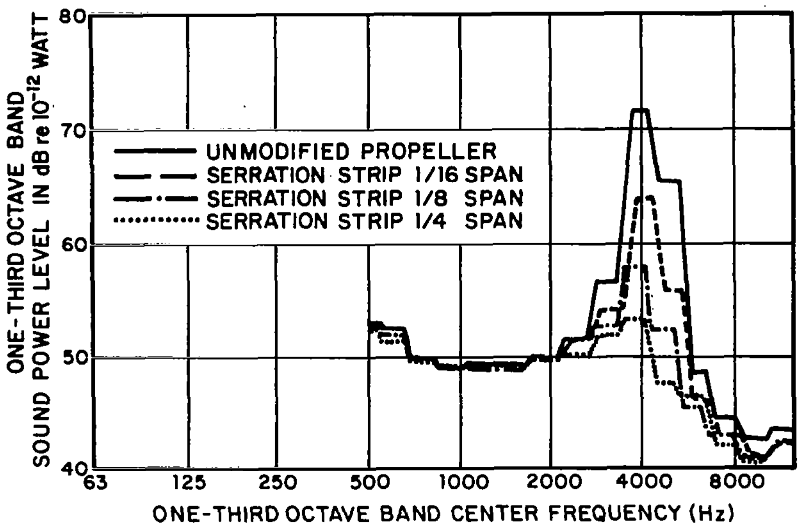


Figure 8.- Effect of properly located leading edge serrations.

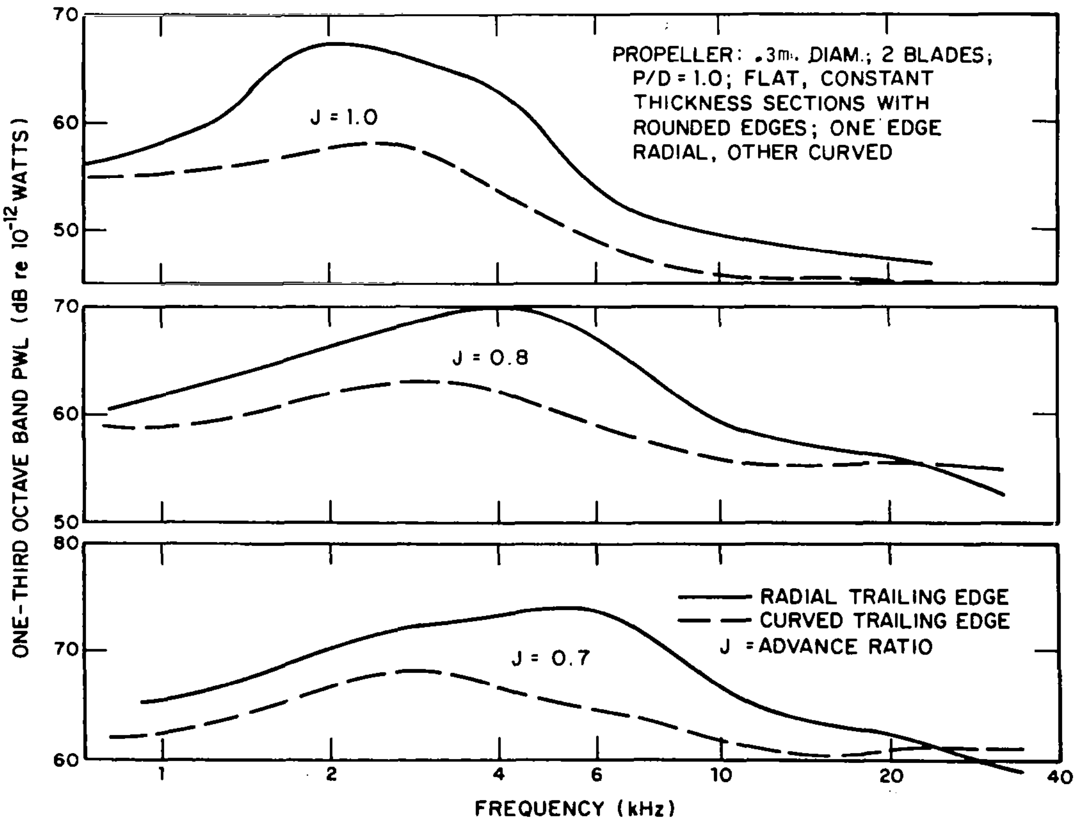
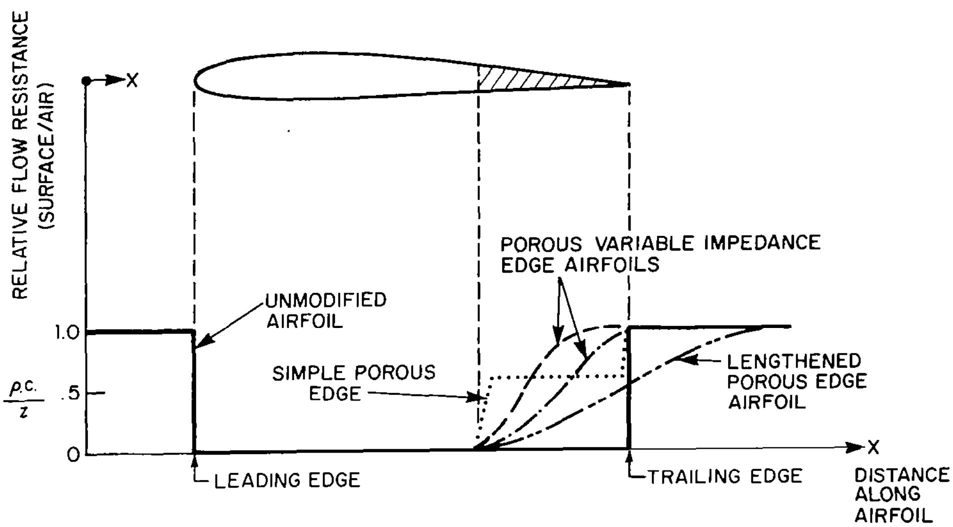
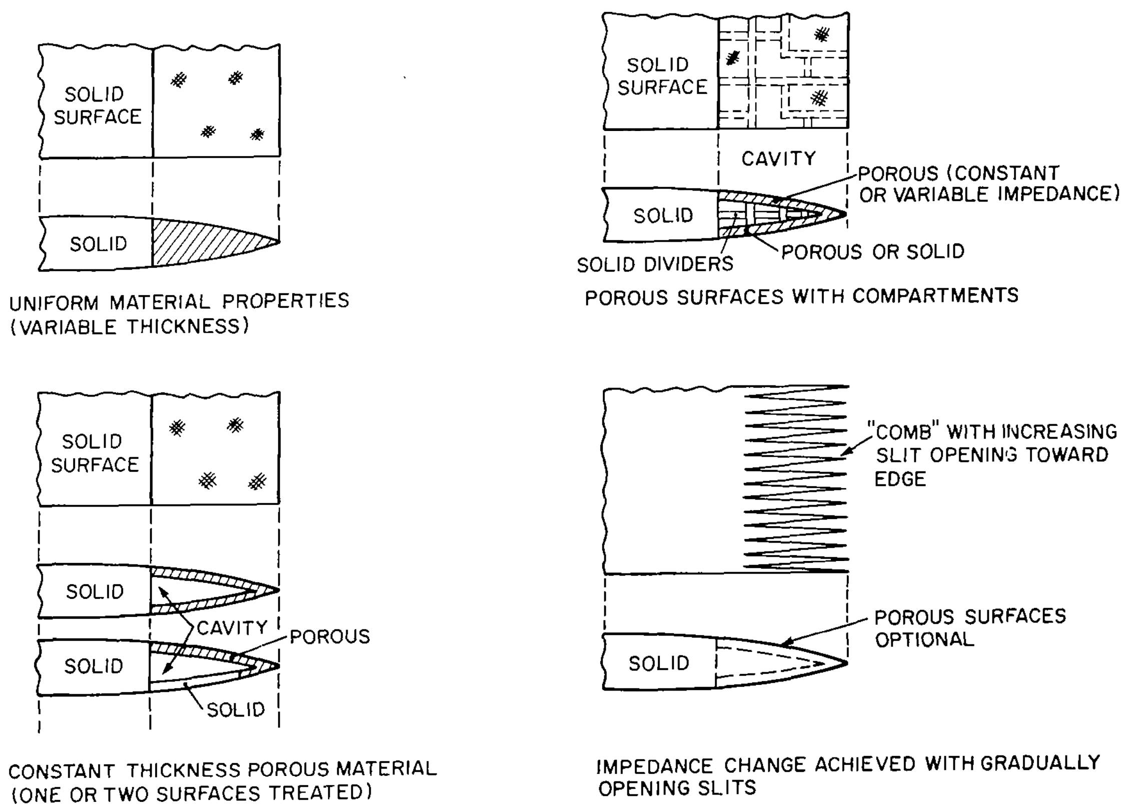


Figure 9.- Effect of sweep on trailing edge noise of rotor.

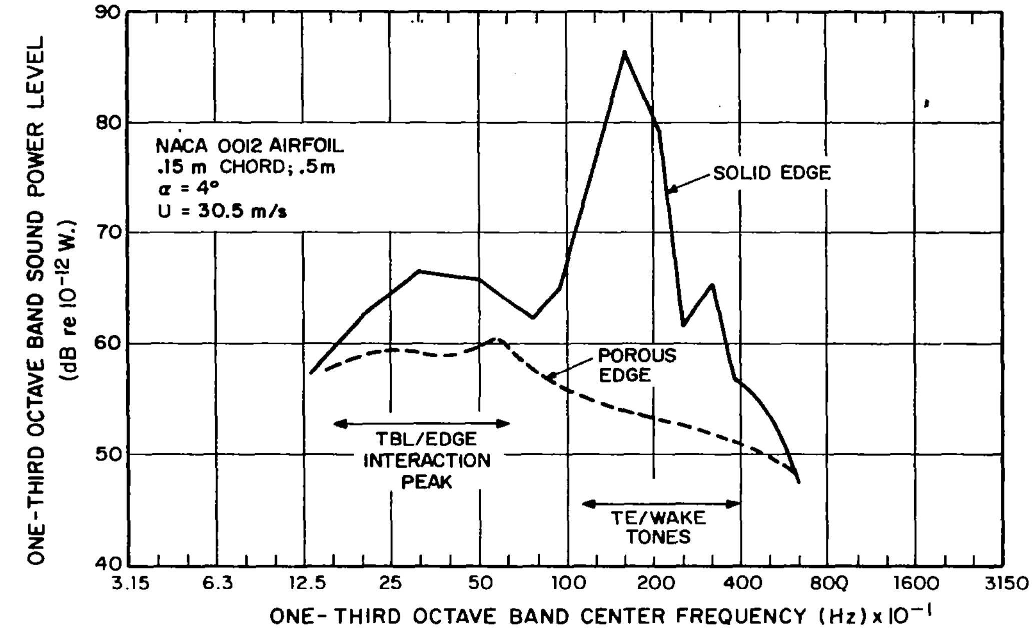


(a) Schematic of porous edge concept and families of impedance changes.

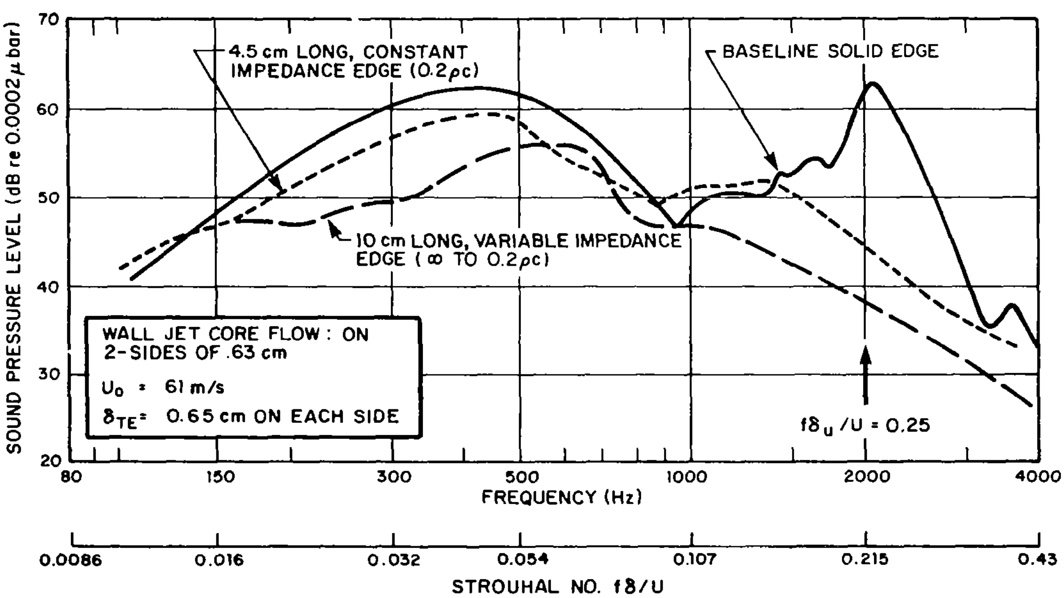


(b) Classes of variable impedance edge structures.

Figure 10.- Porous, or variable impedance, trailing edge concept.



(a) NACA 0012 Airfoil (0.15 m chord; 0.45 m span). U_o = 30 m/s; α = 4^o.



(b) Wall jet/trailing edge noise reduction with two types of porous edges. U_o = 60 m/s.

Figure 11.- Noise reduction of fixed airfoil and plate with variable impedance edges.